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# IonSat: challenging the atmospheric drag with a 6U nanosatellite

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## Abstract

Herein, we present a feasibility study and a mission design for a 6U CubeSat propelled by an iodine NPT-30i miniaturized thruster from the company ThrustMe, to be ready for launch in the early 2020s. The project is led by École polytechnique, Palaiseau, France and supported by the French space agency CNES.

The phase A study shows that the stand-alone propulsion system can be embedded in a 6U CubeSat and be used in the frame of a coherent mission over more than a year, under the condition of integrating a low-requirements payload, mainly in terms of volume. Deployable solar generators, aerobraking strategies and large battery capacities allow the satellite to perform orbit changes and station-keeping at 300km from any orbit inclination. This mission aims to demonstrate a long-duration station-keeping at low altitude and to assess the suitability of electric propulsion devices for tightly constrained missions such as student CubeSats.

As part of the phase A study, the design of an attitude control system relevant to orbit changes is presented, as well as a first thermal analysis of a 6U CubeSat able to generate more than 50W of power. The mission analysis is also explicated: it led to the unusual choice of a spacecraft able to tackle most orbits, at the cost of an adaptation of the mission plan. The general resulting structure and subsystems integration to comply with such constraints is also presented. It is designed to compromise between aerodynamics and total power supply with deployable solar panels and a displaced centre of mass.

This feasibility study tends to show that propulsion for SmallSats is available and has reached milestones in terms of costs and ease of integration that make it compatible with university projects. It is also shown that a 6U CubeSat can today be suitable for a mission needing an electric power subsystem, providing the payload does not have demanding power or volume requirements.

**Keywords:** Student CubeSat, Propulsion, École polytechnique, Electric propulsion

**Acronyms/Abbreviations**

ADCS: Attitude Determination and Control Subsystem
BPS: Bits Per Second
CNES: Centre National d'Études Spatiales
DoD: Depth of Discharge
EIRP: Equivalent Isotropically Radiated Power
GPS: Global Positioning System
Isp: Specific Impulse
LEO: Low-Earth Orbit
OBC: On-Board Computer
OBDH: On-Board Data Handling
PDU: Power Distribution Unit
SSO: Sun-Synchronous Orbit
TC: TeleCommand
TM: TeleMetry
UHF: Ultra High Frequency
VHF: Very High Frequency

## 1. Introduction

Small and microsatellites, now proven to be of great importance for science, telecommunication and many other applications, need an effective propulsion system to unleash their full performances and autonomy. Electric propulsion will provide them the ability to

perform new manoeuvres and control their orbit, especially at low altitude. Possible applications are longer missions at a given altitude and upgraded de-orbiting techniques that would allow new missions while staying compliant with the regulations.
Although current miniaturised electric propulsion systems between 50 and 500W seem compatible with SmallSats of several hundreds of kilograms – the target of many constellation projects now – it is unclear what minimum spacecraft size can be reached while keeping a meaningful mission.
The IonSat mission aims to bring ThrustMe's ion thruster, NPT-30i in low-Earth orbit (LEO), at altitudes ranging as low as 300km. The interests of using the electric propulsion at such altitudes are many. Around 300km, the atmospheric particular density ranges between $2{\cdot}10^{-11}$ kg·m$^{-3}$ and $6{\cdot}10^{-11}$ kg·m$^{-3}$ depending on the solar activity, and the atmospheric drag causes the altitude of a 6U nanosatellite to decrease by 500m per day with deployed solar panels.
Current progress on thruster miniaturization allow CubeSats to feature electric propulsion. Combined with the standardization of the CubeSat technologies, new low-cost missions can be designed. Current work focuses on more ambitious missions, such as interplanetary missions that can now be considered. Propulsion systems give CubeSats a much wider range of missions than previously. Solid propellant provides more safety than gas tanks and allows longer thrust missions, in addition to being more controllable, meaning that cautious manoeuvres can be considered. To assess the new missions that could later be conducted with these new thrust methods, it is necessary to perform in-flight tests. IonSat aims to demonstrate that technological progress has reached a milestone in terms of ease to integrate and work on.

Fig. 1. Artist's view of IonSat orbiting the Earth

## 2. Mission Analysis

### *2.1 Orbit choice*

As defined as its mission, the satellite must perform a descent below 350km (below 300km if the solar activity is favourable, see section *2.3*) and perform an autonomous – as much as possible – orbit control over several months. The initial launch orbit must ensure that this mission is feasible.

- The launch altitude must be low enough so that the descent manoeuvre can be done in a reasonable amount of time and can leave enough propellant for later orbit keeping. With this, the maximum launch altitude is 500km. Such an initial orbit altitude will also be sufficiently low to allow the satellite to de-integrate in the atmosphere in less than 25 years to respect the debris mitigation law, in case of a mission failure at any point of the mission.
- The launch altitude must have an inclination greater than 49° to allow visibility from the École polytechnique ground station located near Paris, France.
- The developing process is willingly short. The target launch window is therefore set to be between 2020 and 2022.

IonSat is aiming at being carried as an additional payload, to minimize costs. To maximise the number of launch opportunities, the mission design focuses on letting as much freedom as possible, in terms of reachable inclinations and sun exposure.

Maximising the available power through the mission drove the analysis to choose a sun-synchronous orbit (SSO). However, results showed that for a 6-18h SSO orbit, though getting 95% of the maximum theoretical power (permanent sun exposure) that allow greater thrust frequency, performing orbit changes has great costs. Indeed, an altitude decrease of 200km requires approximately 200g of propellant due to costly inclination corrections to maintain the sun-synchronism, while IonSat only carries around 400g of fuel. The altitude keeping core mission would therefore be shortened. Additionally, SSO orbit launches are not frequent.

### *2.2 Mission scenario*

Mission analysis therefore led to this mission scenario, with the following phases:

1/ Propelled descent with thrust only
2/ Aerobraking strategy & Thrust
3/ Autonomous altitude keeping, at least 6 months
4/ Additional lower altitude keeping at the end of the mission, provided the fuel provision allows it
5/ De-orbiting

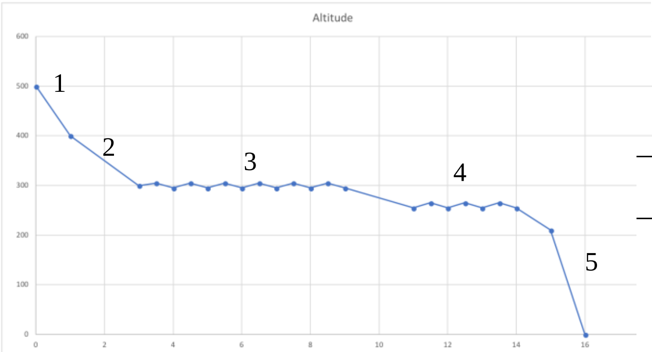

Fig. 2. Mission scenario

The satellite characteristics used in mission analysis calculations stand as follows:

Table 1. Main characteristics of the satellite

| **Characteristic** | **Value** |
|---|---|
| Mass | 10kg |
| Isp | 1200s |
| Thrust | 0.7mN |
| Fuel mass | 400g |
| Maximum drag area | 18dm² |
| Minimum drag area | 2.5dm² |

The ISP, thrust, and fuel mass values are preliminary values taken for the sake of this article after early discussions with the provider, and may not reflect the exact performances of the thruster. This has been taken into account in our study and does not alter our conclusions.

*2.2.1 Descent phase*

With these values, the time required for the descent phase can be calculated. This duration must be set between 3 and 6 months to comply with mission requirements. To control such a short descending duration, IonSat will be using aerobraking: having an orientation with the maximal drag area to oppose particle flux. Geometric drag calculations while using aerobraking with a 18dm² drag area lead to the following values:

Table 2. Altitude loss with respect to the particular density

| **Altitude (particular density)** | **Daily altitude loss** |
|---|---|
| 450km ($1.6 \times 10^{-12}$ kg.m$^{-3}$) | 320m |
| 400km ($3.9 \times 10^{-12}$ kg.m$^{-3}$) | 770m |
| 350km ($9.5 \times 10^{-12}$ kg.m$^{-3}$) | 1800m |

In comparison, the thruster can provide IonSat with a 400m altitude variation per thrust hour.

Combining these previous values with sun exposure durations depending on the orientation, the 3 following descent scenarios from 425 km to 325 km are considered:

Table 3. Descent phase options

| **Option** | **Transfer duration** | **Propellant cost** |
|---|---|---|
| **Stand-alone aerobraking** | 90 days | 0g |
| **Aerobraking + 2 thrust phases/day** | 60 days | 20g |
| **4 thrust phases/day + additional braking *** | 40 days | 40g |

* In this case, the panels face the sun but the satellite keeps losing altitude without aerobraking.

Aerobraking also induces torque on the spacecraft. It can therefore be utilised until the torque generated by the drag and the displaced centre of mass (see section *4*) exceeds the capacity of the magnetorquers.

*2.2.2 Altitude-keeping phase*

On a 50°-inclined orbit, it is always possible to orientate the satellite so that the magnetorquer and the magnetic field form an angle of at least 40°, allowing the magnetorquer to provide 60% of its maximal torque. Selected magnetorquers with a 0.3 A.m$^2$ torque lead to the following values:

Table 4. Use of the magnetorquers

| **Altitude (particular density)** | **Magnetorquers use** |
|---|---|
| 375km ($6.0 \times 10^{-12}$ kg.m$^{-3}$) | 10% |
| 325km ($1.5 \times 10^{-11}$ kg.m$^{-3}$) | 25% |
| 290km ($3.0 \times 10^{-11}$ kg.m$^{-3}$) | 50% |

Aerobraking can therefore not be used when the density of the environment reaches $3.0 \times 10^{-11}$ kg.m$^{-3,}$ because there is a risk of not being able to move the spacecraft back to its nominal attitude. Finally, aerobraking helps for a propellant-economic descent (saving between 20g and 40g of propellant, 5% to 10% of the total initial mass). However, its use is only profitable in the atmosphere areas where the density varies between $3.0 \times 10^{-12}$ kg.m$^{-3}$ and $3.0 \times 10^{-11}$ kg.m$^{-3}$, which is a 130km wide altitude window: most of the transfer.

Considering that 75% of the initial propellant mass remains after the descent phase (1 & 2), the altitude keeping mission (3) duration can be computed. To get these values, average solar activity has been considered, and an average drag area of 3dm².

Table 5. Consequences of the thrust frequency on reachable altitudes and propellant stock

| Thrust 1h… | Reachable altitude (particular density) | Propellant stock |
|---|---|---|
| …every 3 orbits | 260km ($6.1x10^{-11}$ $kg.m^{-3}$) | 1 year |
| …every 4 orbits | 275km ($4.4x10^{-11}$ $kg.m^{-3}$) | 16 months |
| …every 6 orbits | 293km ($2.9x10^{-11}$ $kg.m^{-3}$) | 2 years |
| …every 8 orbits | 306km ($2.2x10^{-11}$ $kg.m^{-3}$) | 33 months |

### *2.3 Note on solar activity*

Solar activity has a great importance on the mission IonSat can achieve. Particular density increases with solar activity at a given altitude. However, solar activity is hard to predict. It follows cycles as shown below.

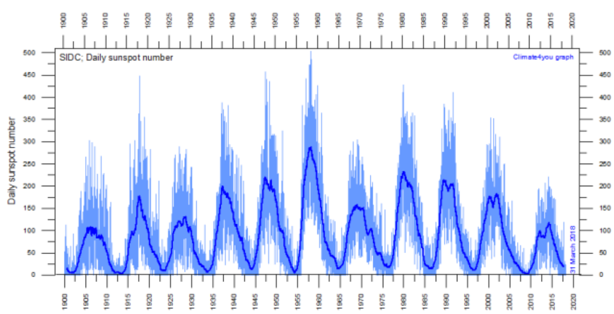


Fig. 3. Evolution of solar activity between 1900 and 2010

The high activity phases have unpredictable durations and intensities. With a launch between 2020 and 2022, IonSat will likely face an increasing solar activity. The altitude values in the previous tables must be corrected with up to date intensity. This only affects the minimum altitude reachable during the mission, which is why particular densities are shown in addition to the altitudes. These are the actual values that were used in the calculations, with altitudes being used for a better understanding.

### *2.4 Sunshine profiles*

The power available in the satellite has consequences on the ADCS hardware choice to perform sun targeting and orientation; on the orbit choices; and most importantly on the achievable thrust frequency, leading to the targeted altitude.

Geometrically, assuming the Earth is a sphere and solar rays come from infinity, eclipses durations can be computed, as shown Fig. 4.

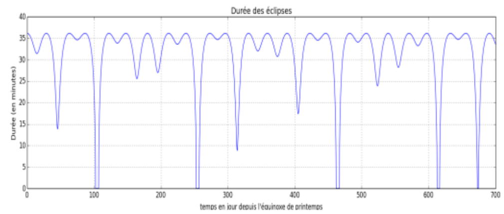

Fig. 4. Eclipses durations per orbit over one year

These durations vary a lot through one year. In some occasions, they are even down to 0 minutes. Most of the time, eclipses last between 30 and 36 minutes, 1/3 of the total orbit period.

An optimal orientation of the satellite during the orbits is required:

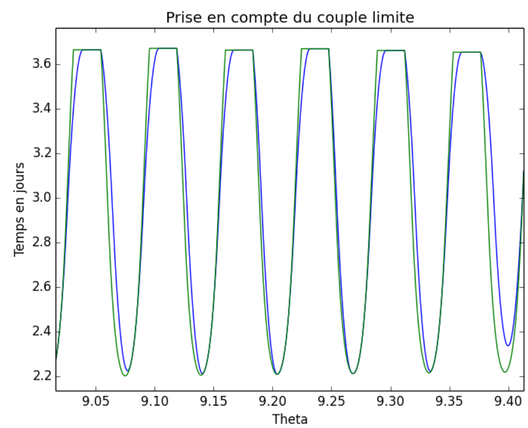


Fig. 5. Optimal and actual orientation of the satellite

On this graph, the blue curve shows the optimal orientation; the green curve shows the actual orientation, taking into account the magnetorquers' limits, hence the satellite slew rate.

The resulting recoverable power is then computed.

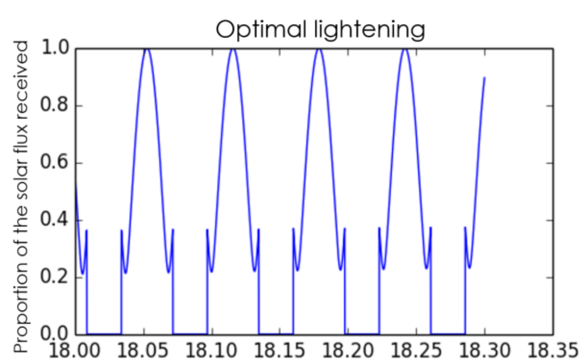


Fig. 6. Recoverable power on different orbits

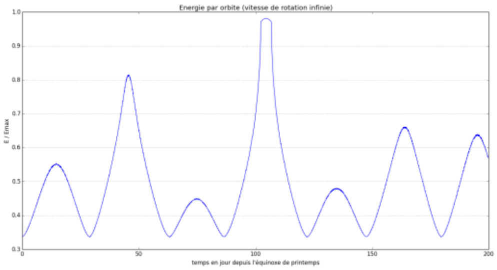


Fig. 7. Daily average recoverable power over a year

Taking into account a pointing error of 5° with respect to the optimal orientation, the worst-case power income is 30% of the best case (perfect sun orientation, no eclipse).

## 3. IonSat power subsystems

Previous analysis in terms of couple shows that an optimised aerodynamism is required to avoid too quick altitude drops. In particular, through altitude keeping phase, the thruster axis will always be parallel to the trajectory, limiting the solar panels orientation to only one degree of freedom. 20% margins have been applied to all the values.

Because the required power of the satellite is more than 60W, the AzurSpace 3g30C solar cells have been chosen, as they are the most efficient. Maximising the density, 50 solar cells can be fitted in the solar panels.

The batteries must comply with the following capacity, mass, operating temperatures and lifetime requirements:

Table 6. Batteries requirements

| Criteria | Level |
|---|---|
| Capacity | > 210Wh |
| Mass | < 4kg |
| Temperature | 5 – 45°C |
| Capacity after 1000 cycles at 30% DoD | > 100Wh |
| Voltage | > 12V |
| Power output | > 60W |

The depth of discharge must never exceed 30% in mission, except for some specific manoeuvres where some extra power might be needed. The chosen batteries are three 144 Wh GomSpace batteries. Combined with the solar cells, the maximum available power reaches 200W.

To assess that the power requirements comply with the mission design, the battery charge cycles are computed, based on 1h thrust phases every 5 orbits.

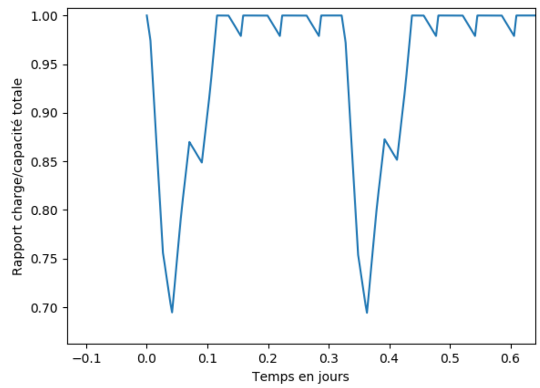


Fig. 8. Depth of discharge of the batteries over 2 cycles

Different voltages are required depending on the subsystems. The chosen Power Distribution Unit (PDU) allows the following electric architecture.

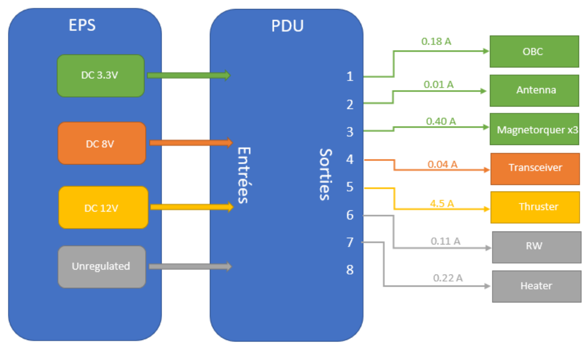


Fig. 9. Electric architecture

## 4. ADCS sizing

While in orbit, the satellite faces different torques. They are listed below.

Table 7. Couples applied on the satellite

| Origin | Torque (Nm) |
|---|---|
| Solar flux | $2\times10^{-7}$ |
| Electromagnetic torque | $3\times10^{-7}$ |
| Gravity gradient | $4\times10^{-7}$ |
| Atmospheric drag | $2\times10^{-6}$ (200 – 400 km) |

The atmospheric drag is computed using the solar radiation, multiplied by the section of the satellite in the worst orientation.

IonSat uses passive stabilisation with a displaced centre of mass a part of its ADCS strategy. Placing the centre of mass 1cm to the front of the satellite, in the thrust direction, generates an additional perturbative torque that can be used to compensate the atmospheric drag. When the atmospheric density is greater than

$5x10^{-12}$ kg.m$^{-3}$, this torque becomes profitable. This is the trigger value for which aerobraking is activated (phase 2). A 1cm offset generates a $6x10^{-7}$ Nm torque.

These computations show that the estimated total torque add up to $C_{dis} = 10^{-6}$ Nm.

During phase 3 (altitude keeping), IonSat is stabilised around 3 axes:

- The thrust axis is always oriented in the direction of the speed vector to minimise drag.
- Rotation around the thrust axis allows solar panels orientation towards the sun.

To ensure stability, the ADCS subsystem must comply with the following requirements:

Table 8. ADCS requirements

| ID | Requirement |
|---|---|
| ADCS-001 | The panels must be oriented towards the sun with a 5° precision. |
| ADCS-002 | The thruster must be aligned with the trajectory with a 5° precision. |

For initial detumbling stabilisation, assuming a 30°/s per axis rotation, the control torque must be greater than $3x10^{-6}$ Nm. Solar panels orientation requires a $10^{-4}$ Nm control torque. It is also necessary to have a $2x10^{-4}$ Nms stored torque. Finally, in the early phase ADCS sizing, $C_{actuators} > 10\ C_{dis}$.

In conclusion, the chosen hardware are magnetometers as captors, magnetorquers and reaction wheels ($C_{rw} = 2.3x10^{-4}$ Nm, $H_{rw} = 1.7x10^{-3}$ Nms). With these, IonSat is compliant in terms of stability and precision of orientation.

To provide more stability, the centre of mass is separated from the geometrical centre. Computations were made to show the stable angle of the satellite with only this passive attitude control, with different atmosphere densitites. A shift of 1mm from the thrust axis has been considered to account for mounting errors. The thruster then tends to push the satellite away from the prograde direction.

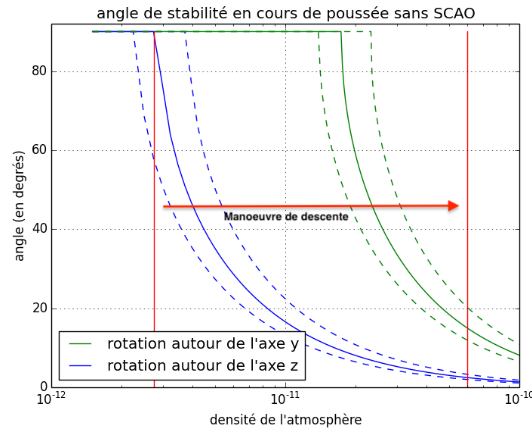


Fig. 10. Stability angle with respect to the atmosphere density

The red vertical lines show the limits in which IonSat will be realising its mission. Regarding the different torques, magnetorquers are maintained, and caution will be put on both the position of the centre of gravity and the attitude control by magnetorquers during assembly and tests.

Because of the thrust phases, contrary to most CubeSats, IonSat needs both attitude control and attitude determination that must interact. Simulations using GMAT show that 1h thrust phases increase the satellite's altitude from 300m on average. The altitude maintaining strategy is therefore to have a thrust cycle, triggered around perigee, when perigee drops below a target altitude that can be set via commands all through the mission. The altitudes of the satellite must therefore be known with a 100m precision (ADCS-005). Additionally, the speed vector drifts from 5° in 75s with the 1mm offset of the centre of mass. Position and speed values must therefore be acquired at least every 30s, for the satellite to react.

This drives the need for an in-board GPS, and an OBC (On-Board Computer) that can compute in real time the constraints on IonSat and activate the actuators. For more safety, the OBDH system must also be updatable in flight in case of an emergency.

## 5. Ground station communication

The main concern is to choose which frequency bands to use for this mission. UHF and VHF bands are used by most radio amateurs, providing a worldwide antenna network. The École polytechnique ground station features a UHF antenna. S bands provide a higher data rate, the antennas are patches that have a directional emission (opposed to the omnidirectional UHF/VHF antennas). Most CubeSats use UHF/VHF transmissions but some typical payloads like cameras require the use of S-band.

École polytechnique's ground station is equipped with two motorised Yagi antennas that can point in the direction of the satellite. It emits in VHF (145.8-146 MHz) with a 1200 BPS rate and receives in UHF (435-438 MHz) with a 9600 BPS rate. Assuming IonSat orbits at a 300km altitude on an ISS-like orbit (meaning with a similar inclination, here set to 51°), the following visible time per orbit is computed:

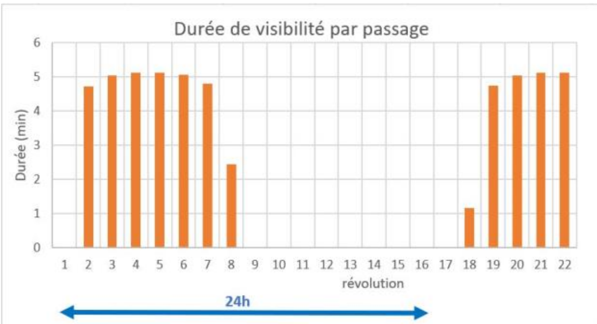

Fig. 11. Visible time over the ground station per orbit

On each revolution, because of the Earth's rotation, the trace on the Earth of the orbit shifts by 22.5°. Therefore, θ changes for every passage. Only 7 passages out of 16 are visible in 24h, with a total daily visible time of about 1939s (32min). TMs and TCs must therefore be carefully planned.

With these characteristics, the maximum transmissible data is 2.33 Mbits/day (uplink, VHF) and 18.6 Mbits/day (downlink, UHF), with the university's ground station. Using a S-band patch provides with approximately 300 Mbits/day data flow.

The downlink must cover housekeeping data such as altitude, temperature measurements, pressure, voltage and current in different subsystems, and thruster information data. It is the second part that conditions the amount of data to be transmitted. It heavily relies on instrumentation on the thruster.

The uplink must cover all TMs such as updated target altitudes, solar panel deployment, survival mode activation. More importantly, mission requirements demand that a software update be possible in case of unexpected misfunctioning, or mission change.

The mission analysis design of the satellite therefore features both UHF/VHF and S-band antennas. The deployable antennas are omnidirectional dipoles that can be used with VHF for receiving TCs and UHF for emitting TMs. For now, the system only exists as a 3U configuration, but it can be mounted on the 2Ux1U side of IonSat, providing the best orientation.

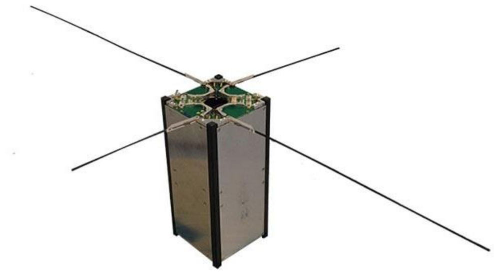

Fig. 12. Example of deployed UHF antennas

The patch S-band antenna has a 1Ux1U surface that can be embedded in the satellite. However, using the S-band requires a new ground station, which can be achieved by either changing the hardware at École polytechnique, or finding partners that let use of their antennas.

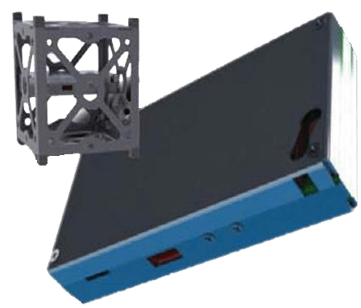

Fig. 13. S-band patch antenna

If the mission baseline is to be updated, the S-band patch can easily be removed.

Link budget has been calculated. The attenuation factors during signal propagations are:

- Joule effect current circulation loss;
- Pointing error with the non-isotropic antennas;
- FreeSpaceLoss, given by $L_{fs} = 32.45 + 20\log(D) + 20\log(f)$, where D is the distant between emitter and receptor and f the signal frequency;
- Atmosphere attenuation, varying with the signal frequency

For practical reasons, the emitted signal power relies on the hypothesis that the emission is isotropic. It is called EIRP (Equivalent Isotropically Radiated Power).

Finally, the received signal power C is given by C = EIRP + Gain – Loss.

| Uplink Budget | | | |
|---|---|---|---|
| Parameters | Values | Units | |
| **G/S : Transmitter** | | | |
| Frequency | 146 | MHz | |
| TX Power Output | 20 | dBW | |
| Line Losses | 1,9 | dB | |
| Transmission Gain | 14 | dBi | Optimal directic |
| EIRP | 32,1 | dBW | |
| **Propagation Losses** | | | |
| Free Space Losses | 137 | dB | |
| Atmosphère Attenuation | 2,1 | dB | |
| Scintillation | 0,7 | dB | |
| Polarisation Losses | 0,2 | dB | |
| Total Losses | 140 | dB | |
| **S/C : Receiver** | | | |
| Antenna Pointing Loss | 0 | dB | mutlidirectionn |
| Antenna Gain | 0 | dB | |
| Line Losses | 1 | dB | majoré |
| Noise Temperature | 313 | K | |
| Noise Power | -161,883644 | dBW | |
| Power Received | -108,9 | dB | |
| Signal to Noise ratio C/N | 52,983644 | | |
| Eb/N0 | 22,1918316 | | |
| **Results** | | | |
| Receiver Sensitivity | -134 | dBW | |
| System Link Margin | 25,1 | dB | |

| Dowlink Budget | | |
|---|---|---|
| **S/C : Transmitter** | | |
| Frequency | 435 | MHz |
| TX Power Output | -3 | dBW |
| Line Losses | 1 | dB |
| Antenna Gain | 0 | dB |
| EIRP | -4 | dBW |
| **Propagation Losses** | | |
| Free Space Losses | 137 | dB |
| Atmosphere Attenuation | 2,1 | dB |
| Scintillation | 0,7 | dB |
| Polarisation Losses | 0,2 | dB |
| Total Losses | 140 | dB |
| **G/S : Receiver** | | |
| Antenna Pointing Loss | 0,5 | dB |
| Antenna Gain | 18,5 | dBi |
| Line Losses | 2 | dB |
| Noise Temperature | 510 | K |
| Noise Power | -161,603357 | |
| Power Received | -128 | |
| Signal to Noise ratio C/N | 33,6033568 | |
| **Results** | | |
| C/N Threshold | 10 | dB |
| System Noise Link Margin | 23,6033568 | dB |

Fig. 14. Link budget

## 6. Thermal analysis

Because IonSat carries a thruster and large batteries, a careful thermal analysis must be made so that all subsystems are constantly powered within the good temperature ranges.

Here are the most critical constraints:

Table 9. Functioning temperature intervals

| Subsystem | Temperature interval |
|---|---|
| Batteries (charge) | 5°C – 45°C |
| Batteries (discharge) | -20 °C – 50°C |

For thermal simulations, Systema-Thermica has been used with a simplified version of the model of the nanosatellite; the material characteristics such as emissivity, absorptivity, thermal capacity; and the orbit characteristics (altitude, inclination). In the model, the thruster is a cube that emits a 10W heat radiation when turned on. The structure is painted with a cold radiative white paint coating (absorptivity: 0.25, emissivity: 0.81). Simulation is done in the Hot Case scenario where the satellite receives the most heat and Cold Case scenario, on 6 orbits with the thruster powered during the whole 2nd orbit.

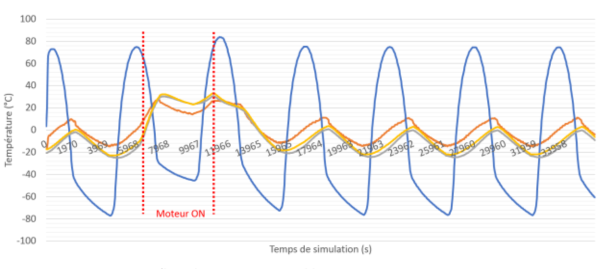


Fig. 15. Temperatures of the subsystems (electronics, structure in red and yellow; solar panels in blue) with radiative paint on the satellite

With the white paint, all subsystems work within their limit range. However, the batteries need active thermal control as the following graphs show:

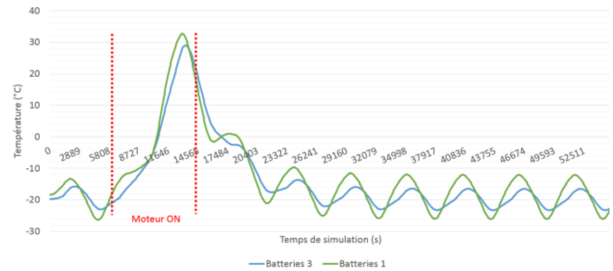


Fig. 16. Batteries temperatures without active thermal control

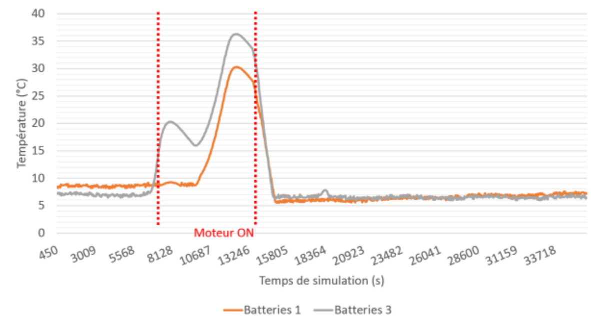


Fig. 17. Batteries temperatures with hot resistors

Hot resistors are added in the second simulation, which shows that the needed power is approximately 3W. The chosen batteries feature an autonomous heating system that can perform this active thermal control. A new simulation will be performed with a more realistic thermal model.

## 7. 6U integration

The selected CubeSat format is a 6U. A 3U would be too small to integrate all components and to get the required power. Thermal requirements would not be matched. Because 6U is also a standard format that can easily be launched with various launch providers, is was selected. Here are some of the main requirements in terms of integration.

Table 10. Structure requirements

| Criteria | Level |
|---|---|
| Dimensions (6U) | 100 x 22.6 x 336 $mm^3$ |
| Total mass | < 12 kg |
| Von Mises Criteria for a 10g acceleration (launch) | Stress < 400 MPa |
| First mode | > 90 Hz |
| Offset of the centre of mass toward the front | To maximise |

| | |
|---|---|
| Distance between the centre of mass and the thrust axis | < 1 mm |

To allow easy mounting and dismounting as well as a compliant structure in terms of vibrations, a "sarcophagus"-type structure has been considered, while waiting for further information on the thruster.

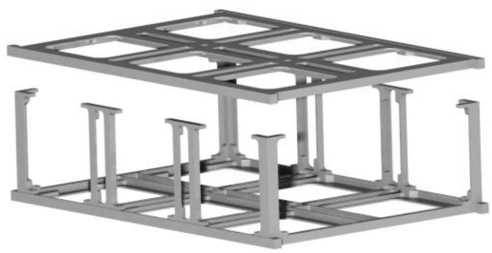

Fig. 18. Selected satellite structure

Access to components is facilitated but producing the pieces can also be complicated. There are in addition less pieces to fix than in a fully assembled structure.

All components must be integrated in this structure, and the centre of mass must be shifted as much as possible. The current full design stands schematically as follows:

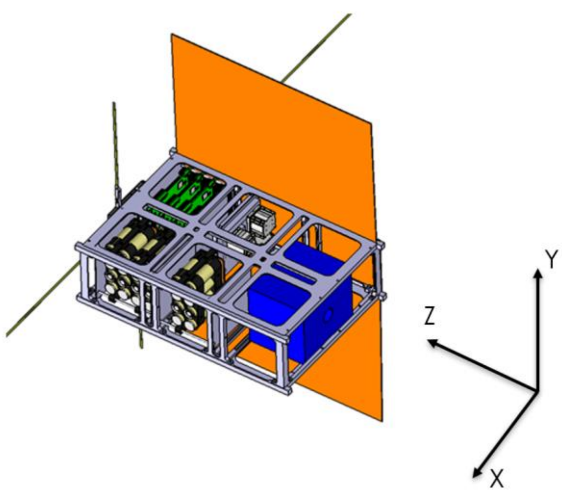


Fig. 19. Components integration in the satellite

Table 11. Centre of gravity shift compared to the centre of mass

| | |
|---|---|
| **X axis** | -9 mm |
| **Y axis** | 0 mm |
| **Z axis** | 20 mm |

Due to the power constraints, IonSat features deployable solar panels to get solar power on the maximum surface. The mechanism for deployment is thermal knives. When the deploying system is triggered, the hinges are released back to their initial form, so that solar panels are deployed.

This structure leads to the following mass and volume budget:

| Component | Mass (grams) | | | Comments |
|---|---|---|---|---|
| | Estimated Mass | Contingency (%) | Estimated mass with margin | |
| | | | | |
| **Structural Subsystem** | | | | |
| *Structure + structure panels* | *1060* | *5* | *1113* | Already flown (5%) |
| *solar panels 2*6U + 3U + 2U* | *960,00* | *5* | *1008* | Already flown (5%) |
| *deployment mechanism* | *100* | *20* | *120* | under development, waiting for additional information |
| *Shielding* | *500* | *20* | *600* | yet to be determined |
| TOTAL Structural Subsystem | | | *2841* | |
| | | | | |
| **Attitude Determination and Control Subsystem** | | | | |
| *Inertial wheels* | *130* | *5* | *136,5* | Already flown (5%) |
| *Magnetorquer* | 196 | *5* | *205,8* | Already flown (5%) |
| *Control card* | 90 | *5* | *94,5* | Already flown (5%) |
| *Gps* | 300 | *20* | *360* | to define |
| TOTAL ADCS | | | *796,8* | |
| | | | | |
| **Electrical Power Subsystem** | | | | |
| *PDU* | *250* | *5* | *262,5* | Already flown (5%) |
| *Batteries* | *1500,00* | *5* | *1575* | Already flown (5%) |
| *Solar cell* | *0,00* | *5* | *0* | included with the solar panels |
| TOTAL Electrical Power Subsystem | | | *1837,5* | |
| | | | | |
| **On-board Computer and On-board Data Handling Subsystem** | | | | |
| OBC | 100 | 5 | *105* | Already flown (5%) |
| TOTAL On-board Computer | | | *105* | |
| | | | | |
| **Communication Subsystem** | | | | |
| UHF/VHF antenna | 80 | 20 | *96* | to confirm, waiting for details from ISIS/Gomspace |
| Communication card | 100 | 5 | *105* | Already flown (5%) |
| TOTAL Communication Subsystem | | | *201* | |
| | | | | |
| **Propulsion** | | | | |
| Thruster + Propellant | 1000 | 20 | *1200* | Under development (20%) |
| TOTAL Payload | | | *1200,00* | |
| | | | | |
| **TOTAL** | | | 6981,30 | |

Fig. 20. Mass and volume budgets

It must be pointed that the satellite currently weighs less than 7kg. It is indeed because the thruster itself is considered as the payload, and there is no additional one. The design lets some space and mass left for an additional payload, but very little power can be used.

## 8. Conclusion

With this full mission analysis and satellite design, it is showed that complex missions using propelled CubeSats can now be achieved, even as university projects. IonSat aims to be one of the lowest operating mission ever launched (the record being GOCE below 300km) and being operational at such low altitudes raises many challenges, especially in terms of torques due to the atmospheric drag. However, with ThrustMe's thruster, it is possible to reach low altitudes, provided the design helps reaching the very high-power requirements. Once again, CubeSats projects show that new low-cost mission can be quickly and efficiently designed. In the future, more ambitious missions can be considered.

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